NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

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A STUDY OF STALL PHENOMENA ON A 45° SWEPT-FORWARD WING

By Gerald M. McCormack and Woodrow L. Cook

Ames Aeronautical Laboratory,
Moffett Field, Calif.

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SUMMARY

An investigation has been made to determine the underlying causes of the undesirable longitudinal characteristics of a 45° swept-forward wing in the moderate and high lift-coefficient range. Three-component force data, pressure-distribution data, tuft studies, and boundary-layer measurements were obtained to enable a detailed correlation between separation phenomena and the longitudinal characteristics of the swept-forward wing.

In the moderate lift range, the occurrence of turbulent separation caused a chordwise redistribution of load over the inboard sections. This, in turn, caused increases in drag and a rearward shift of aerodynamic center but caused no loss of lift. In the high lift range, the occurrence of leading-edge separation caused a loss of section lift that occurred first over the inboard sections and traveled outward as angle of attack was increased. This caused very large increases in drag, a decreased lift-curve slope, and, due to the changes in spanwise loading, caused an extremely large forward shift of aerodynamic center.

In order to improve the longitudinal characteristics of the swept-forward wing, both forms of separation must be postponed. The evidence indicates that effort should be directed first toward postponing leading-edge separation. Only after leading-edge separation is adequately postponed should control of the turbulent boundary layer be attempted.

INTRODUCTION

One of the most difficult problems connected with the design of an airplane employing swept wings is the improvement of the poor characteristics of swept wings in the moderate and high lift-coefficient range. These poor characteristics are partially due to the effects of sweep on the potential-flow field and partially due to the occurrence of separation over the wing. The potential-flow effects are sufficiently well understood and accurate methods of prediction are available (reference 1); consequently, they will not be discussed further herein. The effects of flow separation, on the other hand, are only superficially known and the mechanism of flow separation is quite obscure. Large increases of drag, sudden and large fore and aft shifts of the aerodynamic center, decreases in lift-curve slope, and eventually, of course, establishment of maximum lift coefficient have all been rather generally known to be effects of separation. These effects, furthermore, are manifested at relatively low lift coefficients (sometimes in the lower half of the lift-coefficient range) and thus assume even greater importance than corresponding effects on straight wings (wings with no sweep) where separation is not experienced to any appreciable extent until maximum lift is reached.

To properly approach the problems involved in alleviating the effects of separation, detailed information of the effects of separation is necessary. To obtain this information, a large-scale 45° swept-forward wing was tested in the Ames 40- by 80-foot wind tunnel. This report presents the results of force and pressure-distribution measurements, tuft studies, and boundary-layer measurements made to enable a detailed correlation between separation phenomena and the longitudinal characteristics of the swept-forward wing.

COEFFICIENTS AND SYMBOLS

The data are presented in the form of standard NACA coefficients and symbols as defined in the following tabulation:

C_L lift coefficient $\left(\frac{\text{lift}}{qS}\right)$

C_D drag coefficient $\left(\frac{\text{drag}}{qS}\right)$

C_m pitching-moment coefficient computed about the quarter-chord point of the mean aerodynamic chord $\left(\frac{\text{pitching moment}}{qS\bar{c}}\right)$

c_l section lift coefficient $\left(\frac{1}{c}\int_0^c \text{Pdx} \cos \alpha - \frac{1}{c}\int_0^t \text{Pdz} \sin \alpha \right)$

c.p. center of pressure of section normal force, measured in percent chord aft of the leading edge

P pressure coefficient $\left(\frac{p_l - p}{q} \right)$

p free-stream static pressure, pounds per square foot

p_l local static pressure, pounds per square foot

q free-stream dynamic pressure, pounds per square foot

α angle of attack, degrees

S wing area, square feet

b wing span, feet

$\bar{c}$ mean aerodynamic chord $\left(\frac{\int_0^{b/2} c^2 dy}{\int_0^{b/2} c\ dy} \right)$

c local chord, feet

t maximum thickness of local section, feet

y spanwise coordinate perpendicular to plane of symmetry, feet

x chordwise coordinate parallel to plane of symmetry, feet

z vertical coordinate to airfoil contour perpendicular to chord line, feet

Λ angle of sweep to the quarter-chord line, degrees

δ boundary-layer thickness, feet

MODEL

The geometric characteristics and dimensions of the swept-forward wing are shown in figure 1. The wing had 45° of sweepforward of the quarter-chord line, an aspect ratio of 3.55, a taper ratio of

0.5, no twist, and no dihedral. The wing had an NACA 64A112 section (table I) perpendicular to the quarter-chord line. A photograph of the wing mounted in the tunnel is shown in figure 2.

The pressure orifices were positioned over the upper and lower surfaces of streamwise sections which were located at eight spanwise positions varying from 20.9- to 96.2-percent semispan. The spanwise and chordwise positions of the orifices are listed in table II.

Surveys of the boundary layer were made by means of rakes attached to the wing surface. A typical rake used is shown in figure 3. The rakes consisted of a bank of total-head tubes parallel to the axis of the rake and two banks placed at an angle of 63° to the axis. The tubes parallel to the axis of the rake were used to measure the total-head variation through the boundary layer. The tubes at an angle to the axis were used to determine the variation of flow angle through the boundary layer.

TESTS AND RESULTS

Force and pressure-distribution measurements, tuft studies, and boundary-layer measurements were made through an angle-of-attack range at zero sideslip. Data were obtained at Reynolds numbers from 6.9×10^6 to 14×10^6 (based on the mean aerodynamic chord length of 10.41 ft). However, since these data indicated no appreciable Reynolds number effect, particularly within the purpose of this report, only the data obtained at a Reynolds number of 9.7×10^6 (a tunnel speed of approximately 100 mph) are presented.

Standard tunnel-wall corrections for a straight wing of the same area and span as the swept-forward wing have been applied to angle-of-attack and drag-coefficient data. This procedure was followed, since a brief analysis indicated that tunnel-wall corrections were approximately the same for straight and swept wings of the size under consideration. The corrections applied are as follows:

$$\Delta\alpha = 0.74\ C_L$$

$$\Delta C_D = 0.013\ {C_L}^2$$

The data were corrected for drag tares. Pitching-moment tares were not applied, since they were not known with sufficient accuracy to warrant application. The pitching-moment tares are felt to be quite small, however, and should not appreciably affect the results.

The results of the force tests are shown in figure 4 in the form of conventional lift, drag, and pitching-moment curves. The results of the pressure-distribution measurements are shown in figure 5.

To determine whether or not the pressure-distribution measurements accurately showed all forces acting on the wing, the pressures were mechanically integrated to obtain the total values of lift, drag,[1] and pitching-moment coefficients. The values thus obtained are shown in figure 4 for comparison with the force data. As can be seen, good agreement was obtained and hence it was concluded that the pressure distributions accurately showed the forces acting on the wing.

The boundary-layer measurements consisted of total-head surveys over the wing at various angles of attack. Calibrations of the rakes had indicated that no measurable errors in the total head were incurred until a flow angle greater than 15° was experienced. Therefore, in surveying the boundary layers, data were obtained only at flow angles less than 15°. When flow angles greater than this were encountered the rakes were realined with the flow.

The results of the boundary-layer measurements are shown in figure 6. The boundary-layer thicknesses shown are the heights above the surface of the wing at which the free-stream value of total head was obtained regardless of the direction of flow.

DISCUSSION

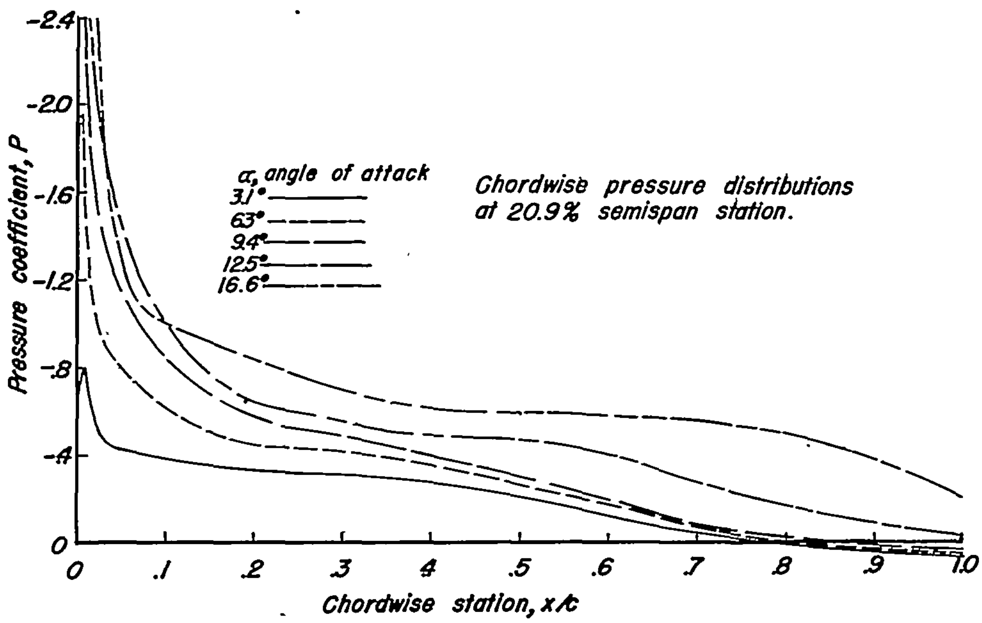
The major influence of separation on the characteristics of swept wings is typified by the longitudinal characteristics of the swept-forward wing which are shown in figure 4. At a moderate angle of attack (starting at about 10°), drag began to rise rapidly and the pitching moments abruptly became more negative (the aerodynamic center shifted aft to 43 percent of the mean aerodynamic chord). At a higher angle of attack (starting at about 15°), drag began to rise even more rapidly, the slope of the lift curve began to decrease, and pitching moments suddenly became positive. (The aerodynamic

[1]Skin-friction drag, naturally, was not indicated by the pressure-distribution measurements. Therefore, to obtain the total-drag coefficients shown in the figure the minimum profile-drag coefficient obtained by the force tests was added to the drag coefficients obtained by integrating the pressure distributions.

center shifted forward to 5 percent of the mean aerodynamic chord forward of the leading edge.) These irregularities, typical of swept-wing characteristics, have generally been attributed to the effects of separation. More exact information of the effects of separation is required, however, to intelligently approach the problems of improving the unsatisfactory characteristics of swept wings operating in the moderate- and high-lift range. In the following sections, pressure distributions, tuft studies, and boundary-layer measurements that were obtained on the 45° swept-forward wing are used to more precisely define the interrelations between separation phenomena and the longitudinal characteristics of the wing. For convenience, the characteristics in the moderate-lift range and the characteristics in the high-lift range are considered separately.

The Moderate-Lift Range

The effects of separation were first evidenced at a lift coefficient of about 0.55 corresponding to an angle of attack of about 10°. Drag began to rise rapidly while pitching moments became abruptly more negative. The reason for the change in the force characteristics can be seen in the pressure distributions over the streamwise section at 20.9-percent semispan, summarized in the following diagram:



These pressure distributions are typical of those obtained over the inboard 41.7 percent of the wing. Up to about 9.4° angle of attack, normal pressure distributions were obtained. At 12.5°, however, pressures failed to recover over the rear portion of the section even though the rate of growth of the forward pressures was little affected.[2] This change in the pressure distributions caused little change in the section lift curves but shifted the center of pressure of the sections rearward. The shift of center of pressure can be seen in figure 7 in which are shown section lift and center-of-pressure curves obtained by mechanically integrating the pressure distributions. The rearward shift of center of pressure occurred at about 10° over the section at 20.9-percent semispan. As angle of attack was increased, sections further outboard exhibited this movement until at about 16°, sections out to 41.7-percent semispan were so affected. This chordwise redistribution of load caused the negative trend in the wing pitching-moment curve between 10° and 16° angle of attack.

The previously mentioned change in the pressure distribution is comparable to changes that occur in pressure distributions over two-dimensional airfoils during the initial stages of turbulent boundary-layer separation. In the two-dimensional case, the failure of the pressures to recover is associated with the formation of a large wake, in effect, an abnormally thick boundary layer following the reversal of flow over the rear portion of airfoil. In the case of the swept-forward wing, however, no reversal of flow was indicated.

An understanding of the phenomena that caused the change in the pressure distributions at about 12.5° angle of attack can be obtained by following the reasoning in reference 2. Separation was taken to mean that the fluid in the boundary layer had lost the component of momentum that carried it across the surface in a direction perpendicular to the long axis of the wing. Therefore, when separation occurs over an oblique wing it was reasoned that the boundary layer would flow in a direction parallel to the long axis of the wing and on this basis would not necessarily be expected to be accompanied by a reversal of the boundary-layer flow. In this respect, separation over an oblique wing would differ from separation over a two-dimensional section. A rapid increase in boundary-layer thickness would, however, be expected due to the combined effects of chordwise and spanwise flow. In this respect, separation over an oblique wing would be similar to separation over a two-dimensional section. Boundary-layer measurements substantiated the above

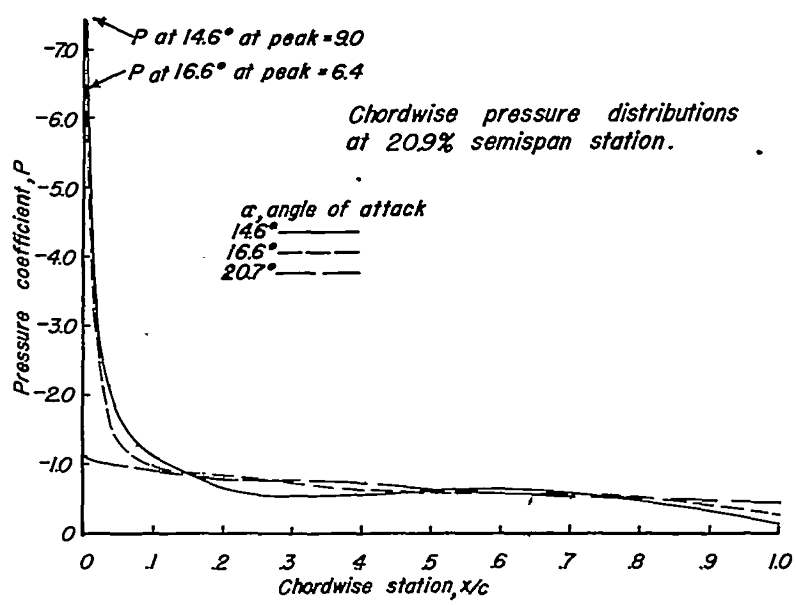
[2]Since pressure distributions were not obtained between 9.4° and 12.5° angle of attack, this change in the pressure distributions could have occurred at any point between those two angles.

reasoning. In figure 6, it can be seen that over the streamwise section at 20.9-percent semispan the boundary layer thickened rapidly at 12.5° angle of attack. This is the angle of attack at which the pressures first failed to recover and at which tufts began to oscillate but did not reverse direction. Furthermore, the portion of the chord over which the boundary layer thickened was the same as that over which the pressures failed to recover.

The effects of separation of the turbulent boundary layer were evidenced both in thickened boundary layers and in a failure of pressures to recover over the rear portion of the sections even though reversal of flow did not occur. Thus it apparently is a form of turbulent separation that caused the initial increase in drag and the rearward shift of aerodynamic center.

High-Lift Range

Above 16.6° angle of attack, the force tests showed that the negative trend of pitching moments was rapidly and completely reversed, with the result that marked longitudinal instability was indicated. This was accompanied by greatly increased drag and a gradual decrease in lift-curve slope. Corresponding changes that took place in the pressure distributions over the streamwise section at 20.9-percent semispan can be seen in the following diagram:



These pressure distributions are typical of those obtained at the higher angles of attack over all except the tip sections of the wing. It will be observed that above 16.6° the suction pressures over the leading edge began to decrease. This caused a loss of lift over the section when the angle of attack was further increased. This loss of lift occurred first over the inboard sections and, as angle of attack was further increased, occurred over sections farther outboard. For example (fig. 7), loss of lift occurred at the 20.9-percent semispan section at about 16.6° angle of attack, had progressed out to 41.7-percent semispan at 20° angle of attack, and did not occur at 80-percent semispan until about 30° angle of attack. The loss of lift over the inboard sections caused a change in the spanwise loading in which the spanwise center of load was shifted outward. The outward movement of the center of load, due to the forward sweep of the wing, caused positive pitching moments. As a result the aerodynamic center moved forward to 5 percent of the mean aerodynamic chord forward of the leading edge.

The loss of the leading-edge suction peak evidently is the result of a permanent separation of the laminar boundary layer at the airfoil nose with no subsequent reattachment. It is apparently largely independent of the turbulent separation that occurs over the rear portion of the airfoil. This is evidenced by its sudden appearance and its rapid spread beyond the area affected by turbulent separation. The boundary-layer measurements were not sufficiently detailed to completely verify the foregoing inferences. The evidence strongly indicates, however, that the greatly increased drag, the decreased lift-curve slope, and the forward shift of aerodynamic center were caused primarily by a leading-edge type of separation.

The section lift characteristics (fig. 7) show the influence of the spanwise boundary-layer drain over the swept-forward wing. The maximum lift coefficients of sections perpendicular to the quarter-chord line varied from 1.01 at 28.1-percent semispan to 1.5 at 71.4-percent semispan. Two-dimensional data for the airfoil section used (NACA 64_1A112 perpendicular to the quarter-chord line) show that a maximum lift coefficient of 1.5 is attainable in a comparable range of Reynolds number. In comparing two-dimensional and three-dimensional values, however, account must be taken of the effects of wing sweep. If the lift coefficients shown in figure 7 had been based on the velocity component perpendicular to the leading edge in accordance with the concepts of simple sweep theory (reference 3), section maximum lift would vary from 2.02 at 28.1-percent semispan to 3.0 at 71.4-percent semispan. On this basis, the sections attained considerably higher maximum lift coefficients than are attainable in two-dimensional flow. From this it can be concluded that insofar as maximum lift is concerned boundary-layer effects resulting from wing sweep are not detrimental. It must be noted that the application of simple sweep theory in the above manner is not intended as a precise correction, but is used only to enable an approximate comparison between two-dimensional and three-dimensional values.

Alleviation of Separation Effects

The problems involved in alleviating the poor characteristics of the swept-forward wing are evident from the foregoing discussion. These consist of a postponement of turbulent separation in the moderate lift range and a postponement of leading-edge separation in the high-lift range. Both postponements, of course, should be to an angle of attack at least as high as the maximum that can be encountered in flight. Beyond the flight range of angle of attack the stall progression must be such that no longitudinal instability results, since instability would possibly curtail the usable lift range.

Of the two types of separation encountered, effort should first be directed toward delaying leading-edge separation. There are several reasons for this. The range over which leading-edge separation must be delayed is fairly small, from 16° to somewhere in the neighborhood of, say, 20° (a possible maximum ground angle). Hence, simple mechanical nose modifications, such as a plain leading-edge flap or a Kruger flap, should provide adequate control. Furthermore, any beneficial changes in leading-edge flow will be reflected as beneficial changes in trailing-edge flow and consequently delays in turbulent separation should result. On the other hand, the influence of trailing-edge flow on leading-edge separation is uncertain, and any benefits obtained by attempting to control turbulent separation first might soon be overshadowed by the detrimental effects of leading-edge separation.

Application to General Case of Swept Wings

The present investigation was concerned primarily with a particular configuration of a swept-forward wing. However, if reasonable consideration is given to the effects of physical changes, certain inferences can be drawn as to the behavior of other swept wings whether swept forward or swept back. The effects of separation and section stall on the characteristics of swept-back wings should be quite similar to the corresponding effects on the swept-forward wing. For a swept-back wing with like airfoil sections the first occurrence of separation might be expected to be turbulent separation over the outboard area. Again the center of pressure of the sections affected would be shifted aft resulting in more negative pitching moments. Turbulent separation would again be expected to be followed by leading-edge separation. Loss of lift due to leading-edge separation will occur first at the outboard area and, as angle of attack is further increased, will occur over sections farther inboard. Thus, loss of section lift will travel forward relative to the moment center and a tendency toward longitudinal instability will result. On this basis, alleviation of the poor characteristics of swept-back wings can be approached along the same line as previously described for the swept-forward wing.

CONCLUDING REMARKS

Tests made on a 45° swept-forward wing showed the flow conditions underlying the poor longitudinal characteristics of the wing in the moderate- and high-lift range.

In the moderate-lift range (C_L = 0.5 to 0.7), the occurrence of turbulent separation caused a chordwise redistribution of load over the inboard sections. This caused increases in drag and a rearward shift of the aerodynamic center (from $0.26\bar{c}$ to $0.43\bar{c}$) but caused no loss of lift.

In the high-lift range (C_L = 0.7 to 1.04), the occurrence of of leading-edge separation caused a loss of section lift that occurred first over the inboard sections and traveled outward as angle of attack was increased. This caused very large increases in drag, a decreased lift-curve slope, and, due to the changes in spanwise loading, caused an extremely large forward shift of aerodynamic center (from $0.43\bar{c}$ to $0.05\bar{c}$ forward of the leading edge).

In order to improve the longitudinal characteristics of the swept-forward wing, both forms of separation must be postponed. The evidence indicates that effort should be directed first toward postponing leading-edge separation. Only after leading-edge separation is adequately postponed should control of the turbulent boundary layer be attempted.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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2. Jones, Robert T.: Effect of Sweepback on Boundary Layer and Separation. NACA TN No. 1402, 1947.

3. Betz, A.: Applied Airfoil Theory. Unsymmetrical and Non-Steady Types of Motion. Vol. IV of Aerodynamic Theory, div. J, ch. IV, sec. 4, W. F. Durand, ed., Julius Springer (Berlin), 1935, pp. 94 – 107.

TABLE I

ORDINATES OF NACA 64A112 a=0.8 (MODIFIED) AIRFOIL SECTION
[Stations and ordinates given in percent of airfoil chord]

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.454	.988	.988	-.932
.699	1.197	.801	-1.117
1.192	1.523	1.308	-1.403
2.433	2,123	2.567	-1.911
4.924	2.967	5.076	-2,607
7.421	3.606	7.579	-3.120
9.921	4.136	10,079	-3.540
14,924	4.969	15.076	-4.189
19.931	5.597	20,069	-4.667
24,940	6.060	25.060	-5.008
29,950	6.383	30.050	-5.235
34.961	6.577	35.039	-5.353
39.973	6.632	40.027	-5.354
44.985	6.520	45.015	-5.206
44.997	6.270	50.003	-4.940
55.007	5.907	54,993	-4.581
60.017	5.452	59.983	-4.150
65.025	4.916	64.975	-3.662
70.032	4.312	69.968	-3.130
75.038	3.658	74.962	-2,578
80.045	2.967	79.955	-2.033
85.044	2.242	84.956	-1.520
90.031	1.508	89.969	-1.018
95.016	.767	94.984	-.521
100.000	0	100.000	0

L.E. radius: 0.994
Slope of radius through L.E.: 0.0475

NACA

TABLE II

LOCATION OF PRESSURE ORIFICES

Spanwise Positions[1] of Orifices	
Station No.	Percent Semispan
1	20.9
2	28.1
3	41.7
4	57.4
5	71.4
6	85.0
7	92.5
8	96.2

Chordwise Positions[2] of Orifices (on Upper and Lower Surfaces at Each Station[3])	
Orifice No.	Percent Chord
0	0
1	.25
2	.50
3	1.0
4	1.5
5	2.5
6	3.5
7	5.0
8	7.5
9	10.0
10	20.0
11	30.0
12	40.0
13	50.0
14	60.0
15	70.0
16	80.0
17	90.0
18	97.5

[1]Spanwise positions are measured perpendicular to the plane of symmetry.

[2]Chordwise positions are measured in percent of the windstream chord.

[3]On station 8, orifices no. 1, 2, 3, 4, 6, 8, 11, 13, 15, 17, and 18 were omitted.

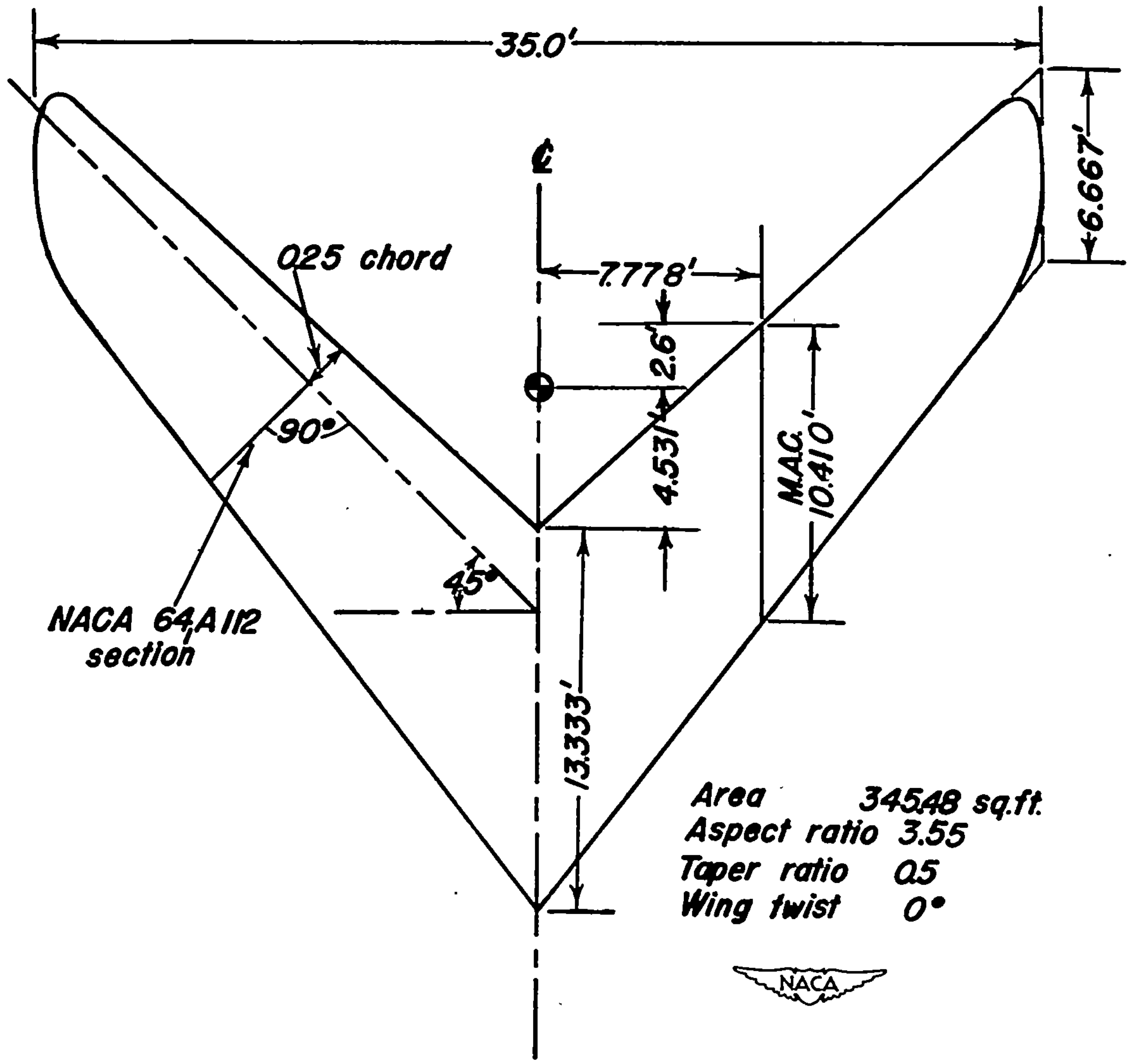


Figure 1.- Geometric characteristics of 45° swept-forward wing.

Figure 2.- The 45° swept-forward wing mounted in the 40- by 80-foot wind tunnel.

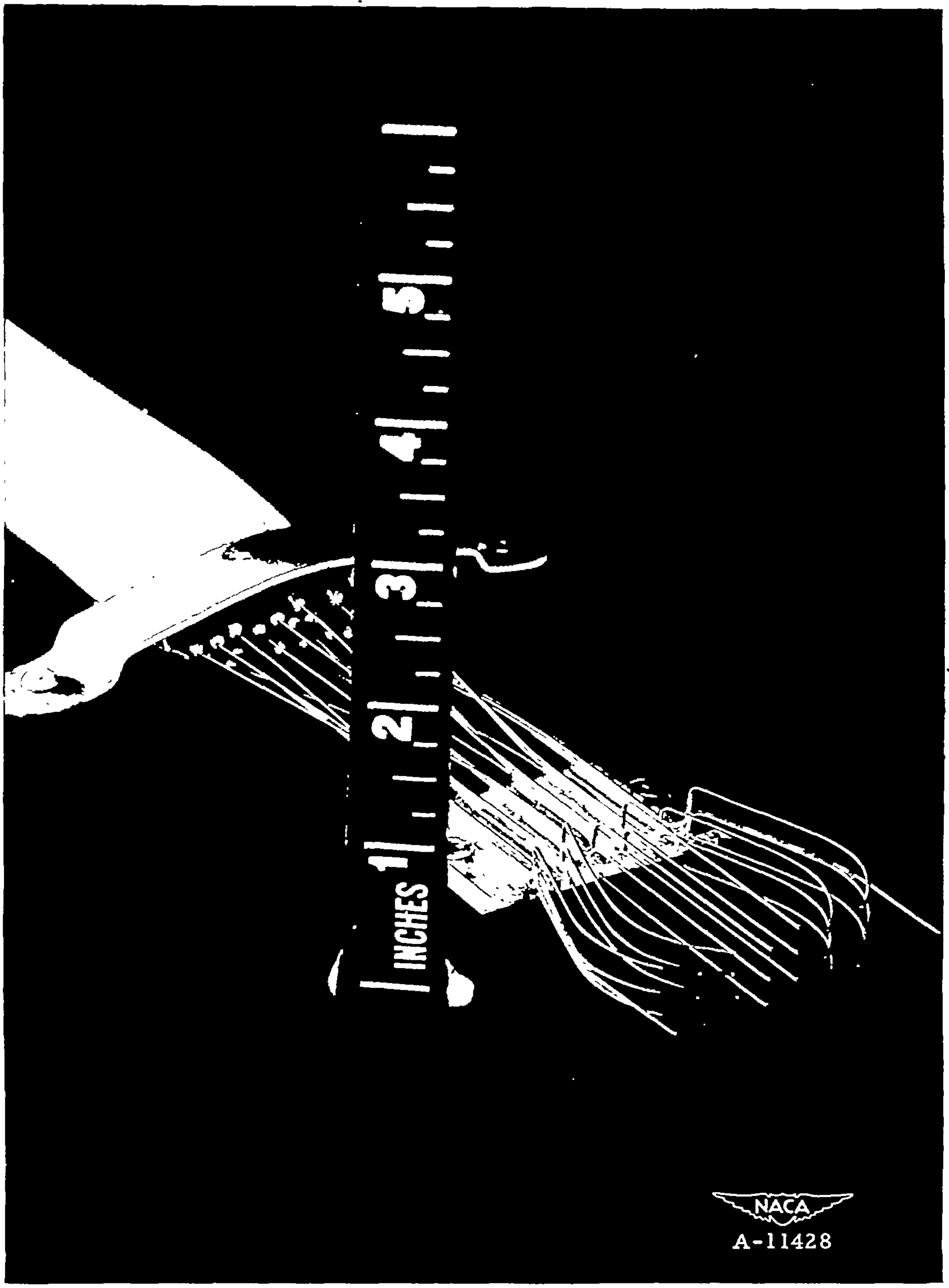


Figure 3.— A typical rake used to survey boundary layers over 45° swept-forward wing.

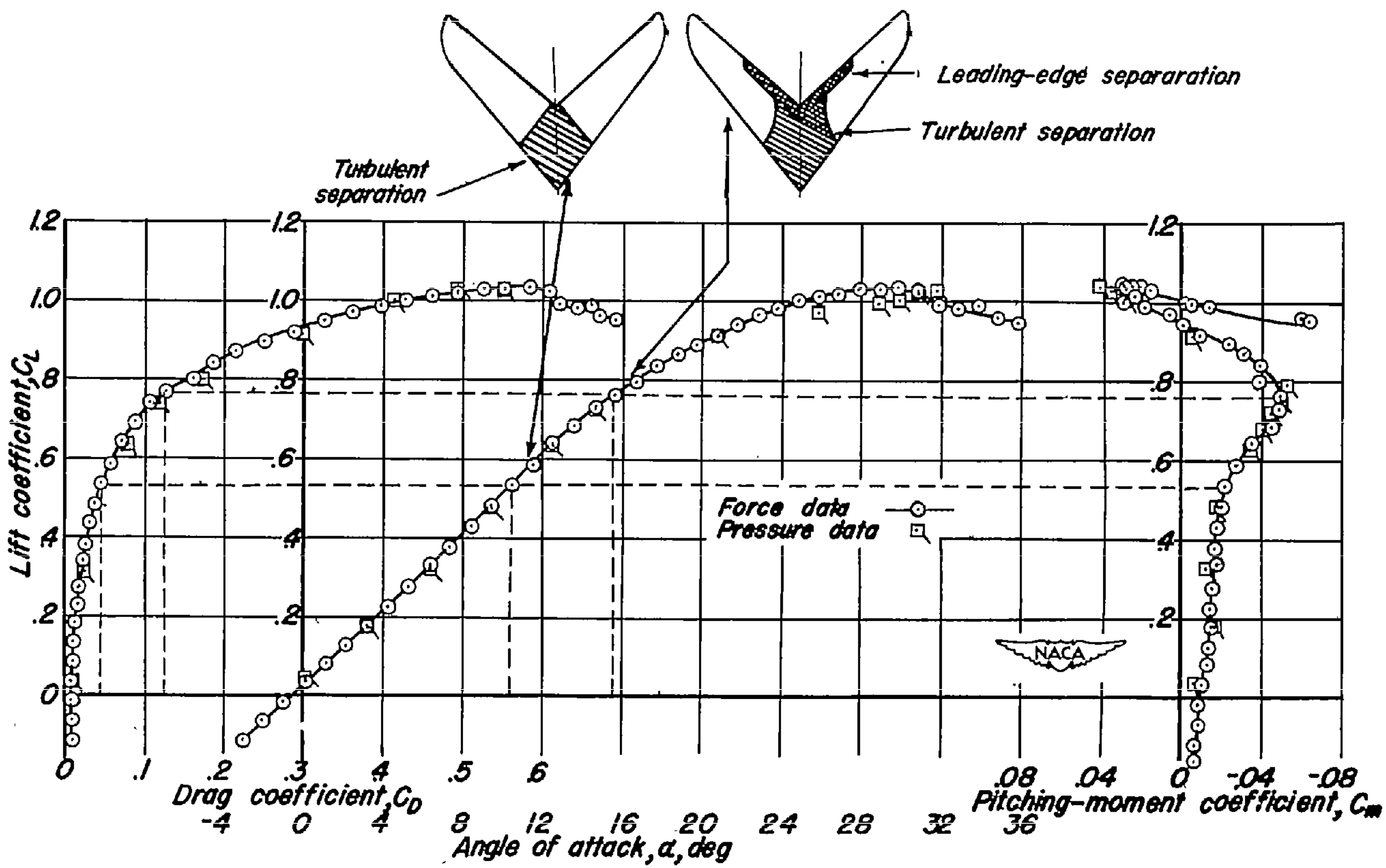
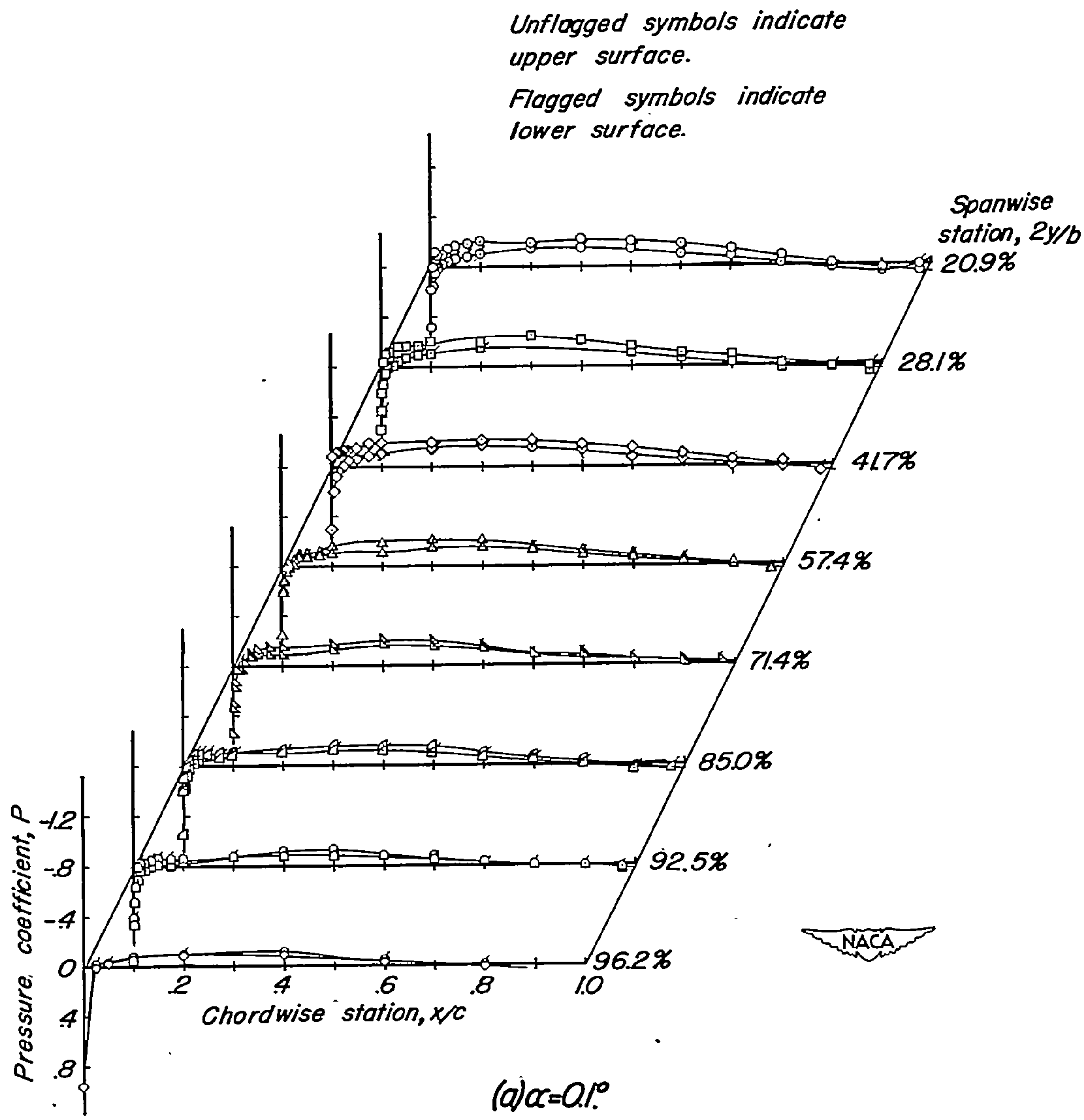
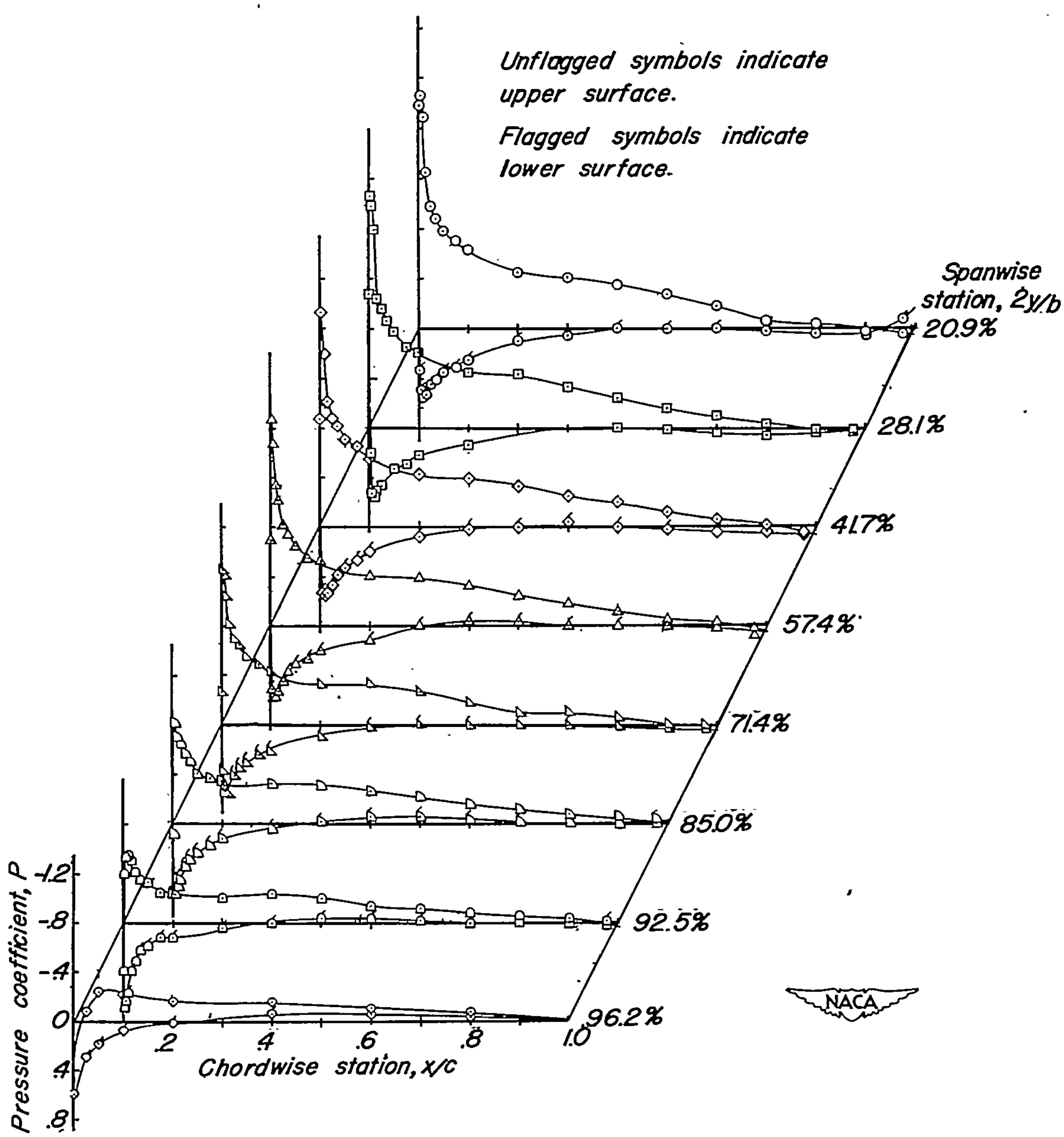


Figure 4.- Longitudinal characteristics of 45° swept-forward wing.



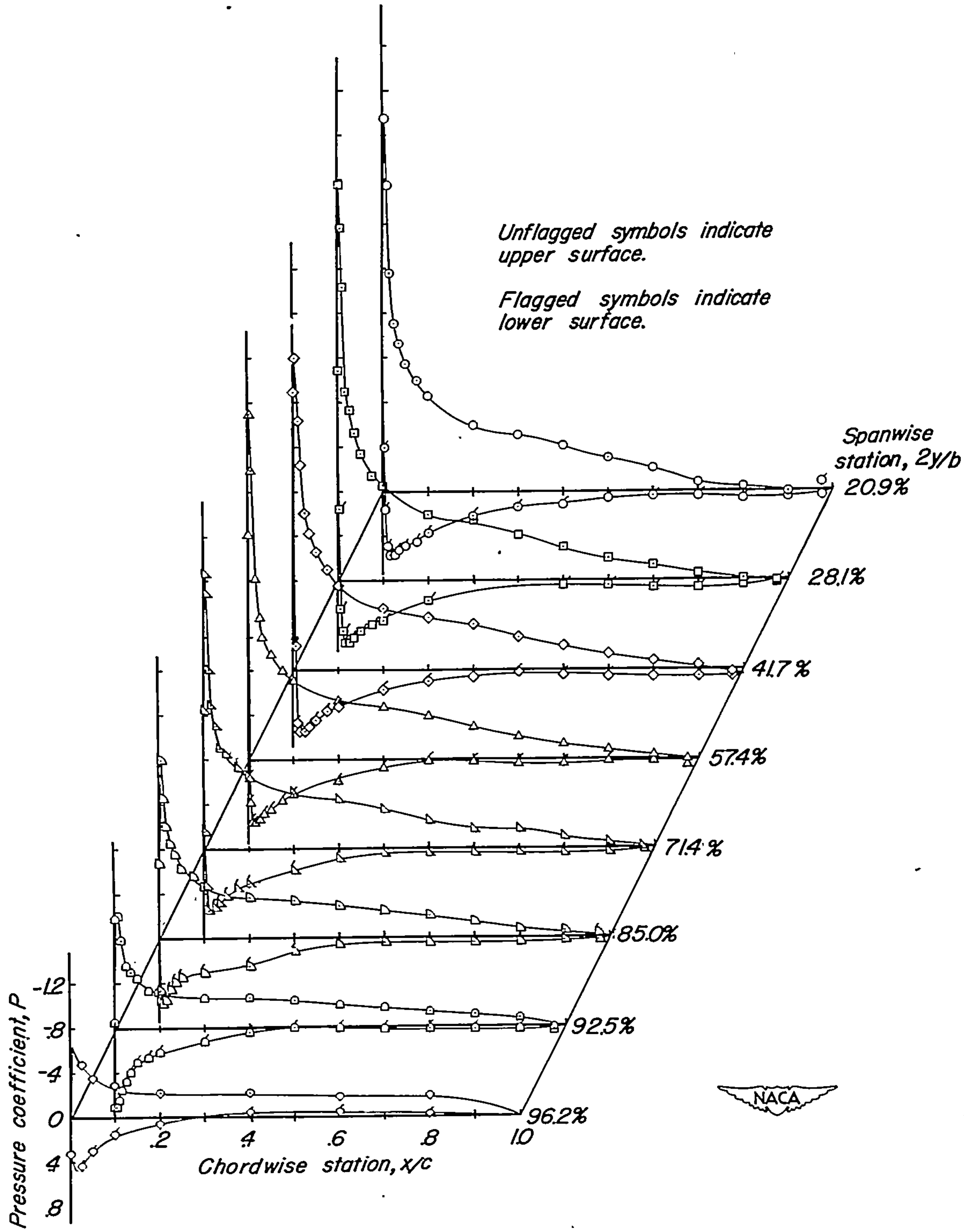
(a) $\alpha = 0.1°$

Figure 5.—Chordwise pressure distributions for 45° swept-forward wing.



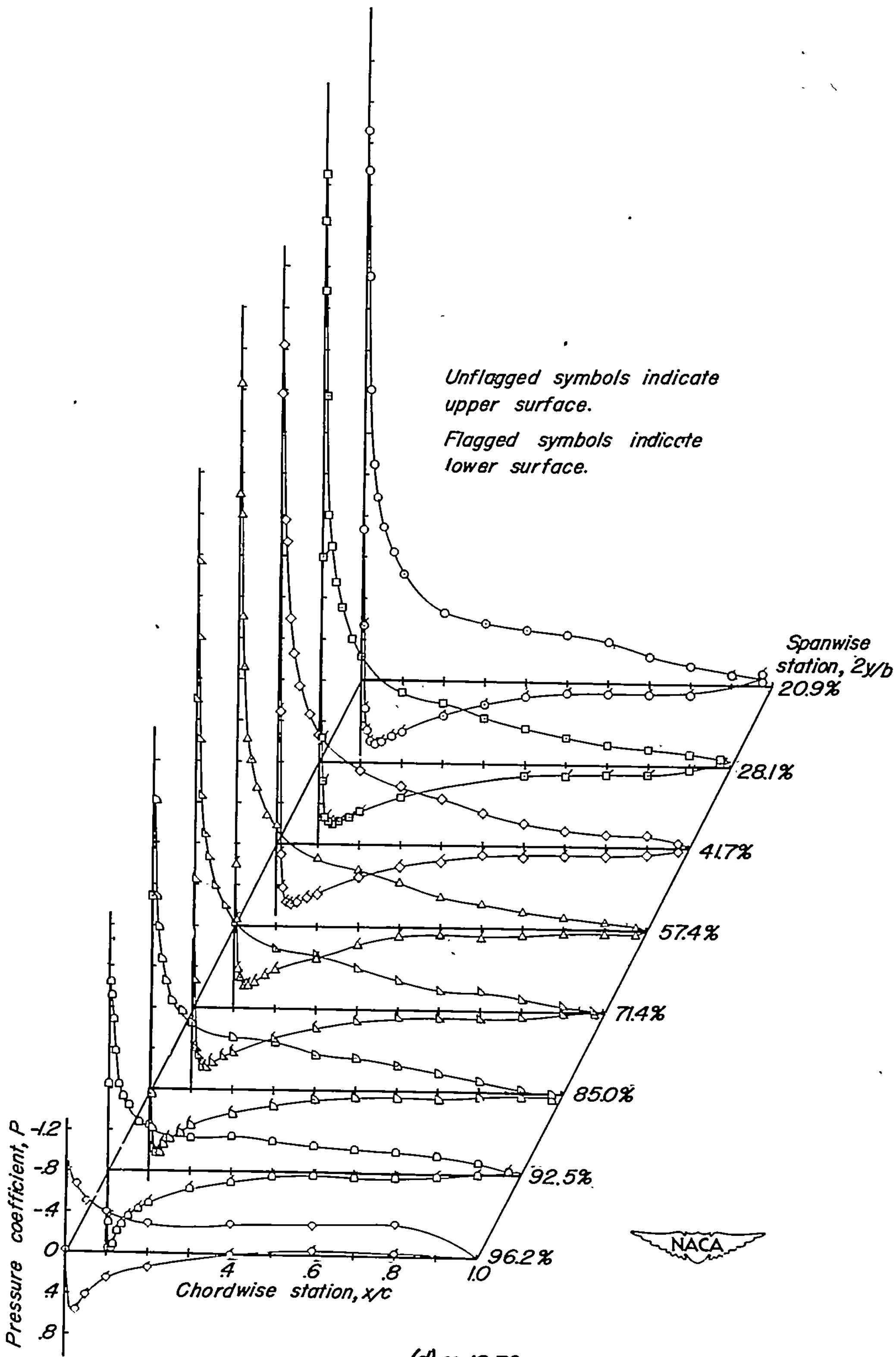
(b) $\alpha = 63^{\circ}$

Figure 5.—Continued.



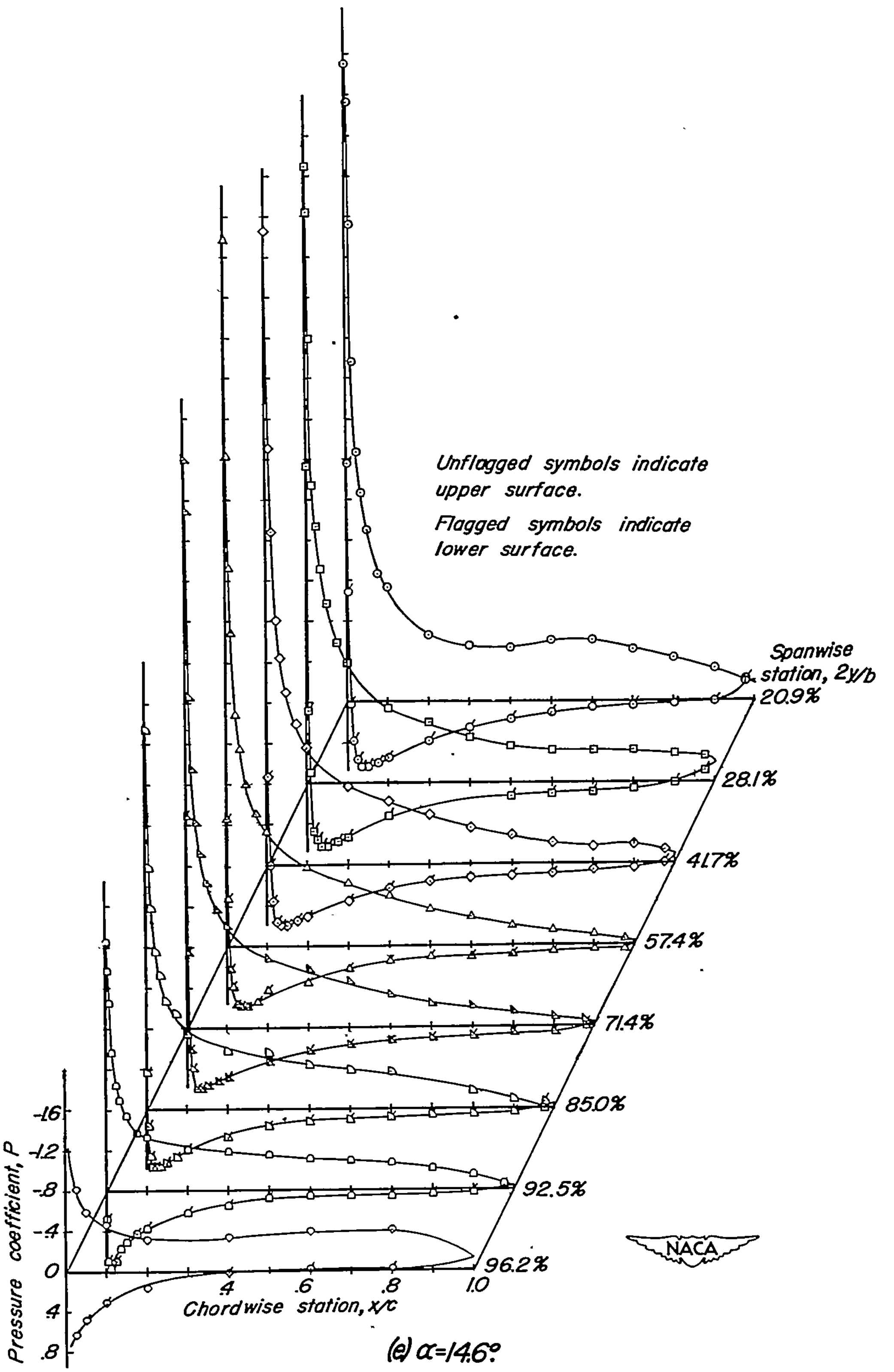
(c) α=9.4°.

Figure 5.—Continued.



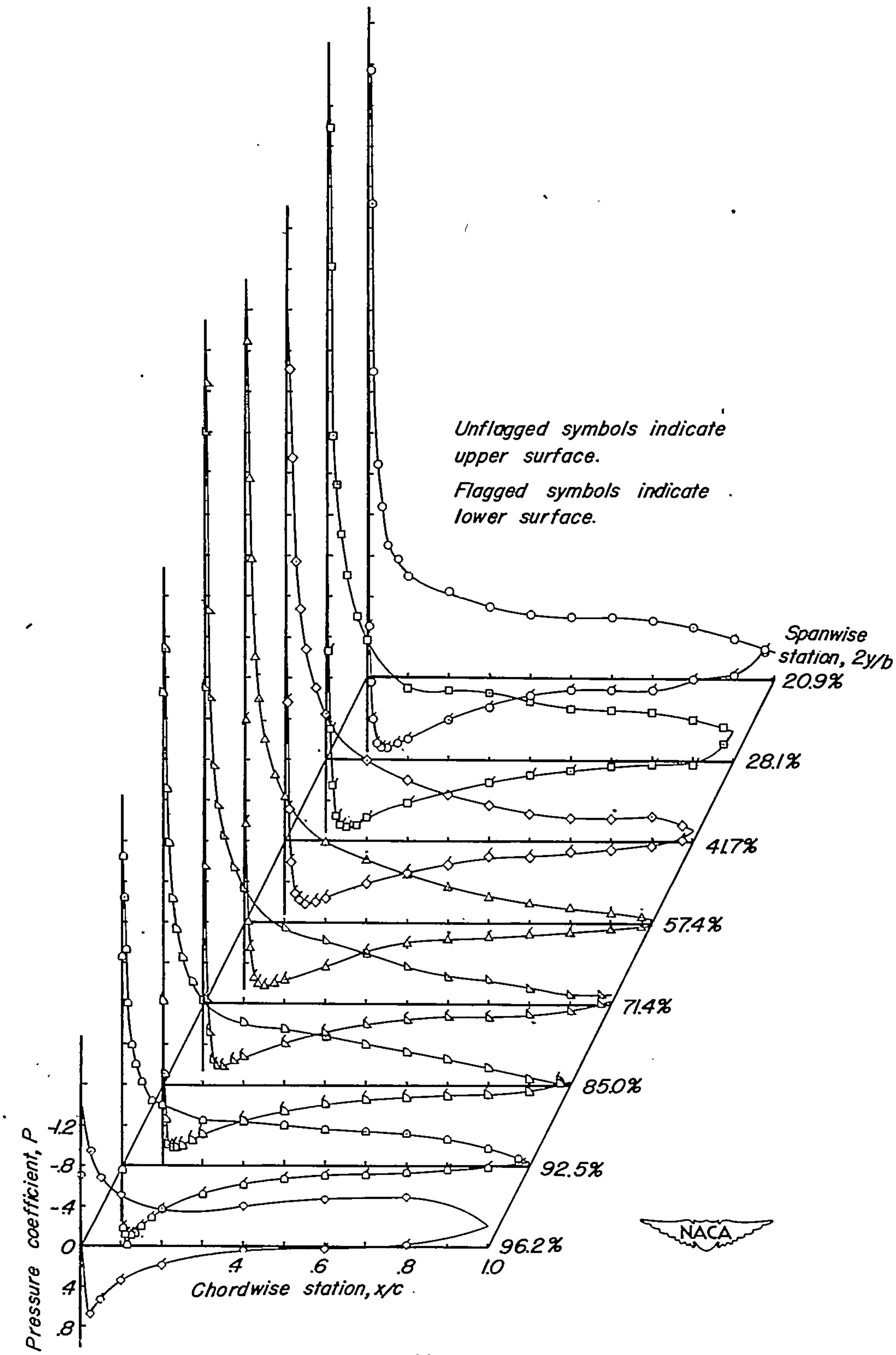
(d) α=12.5°

Figure 5.– Continued.



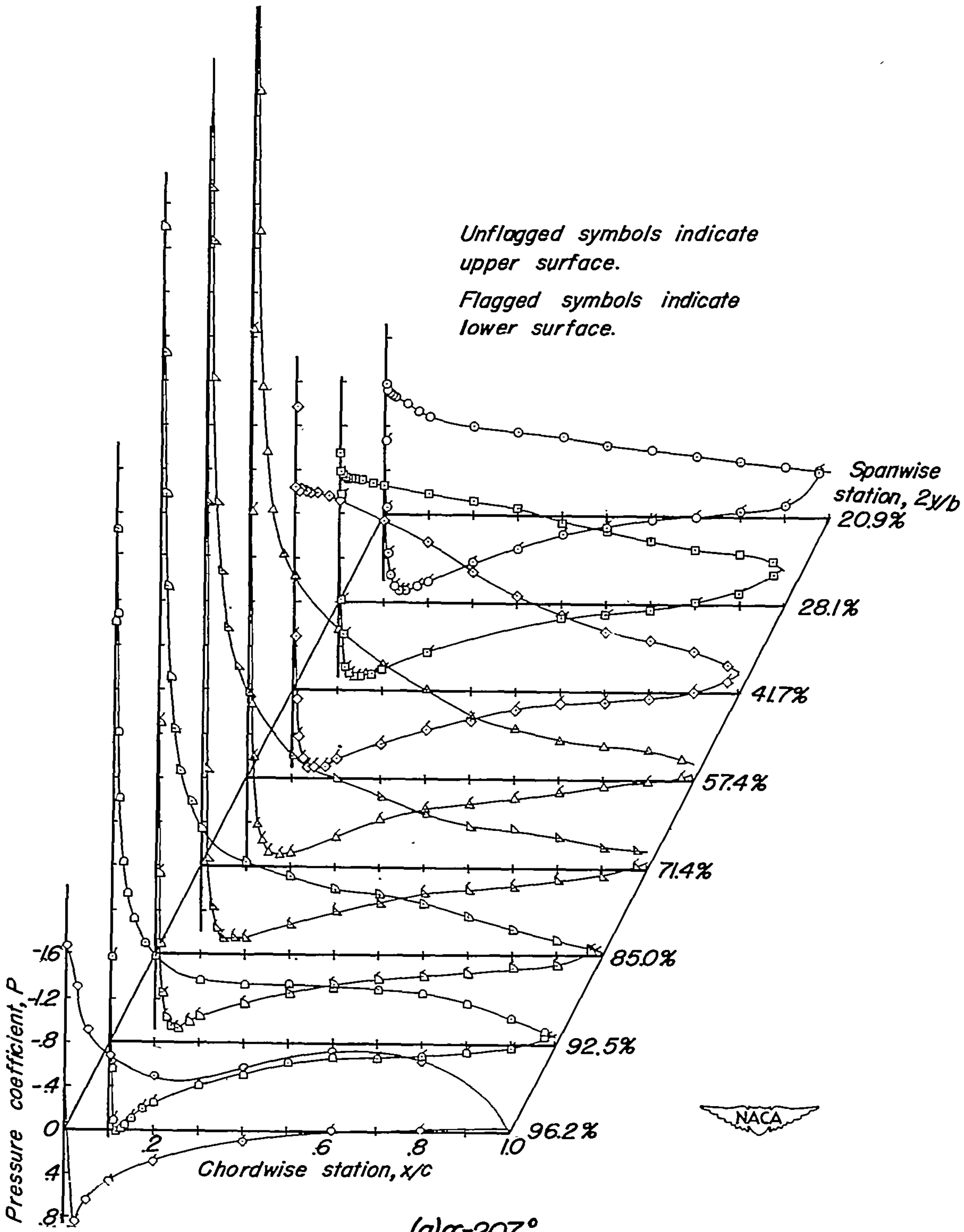
(e) α=14.6°

Figure 5.– Continued.



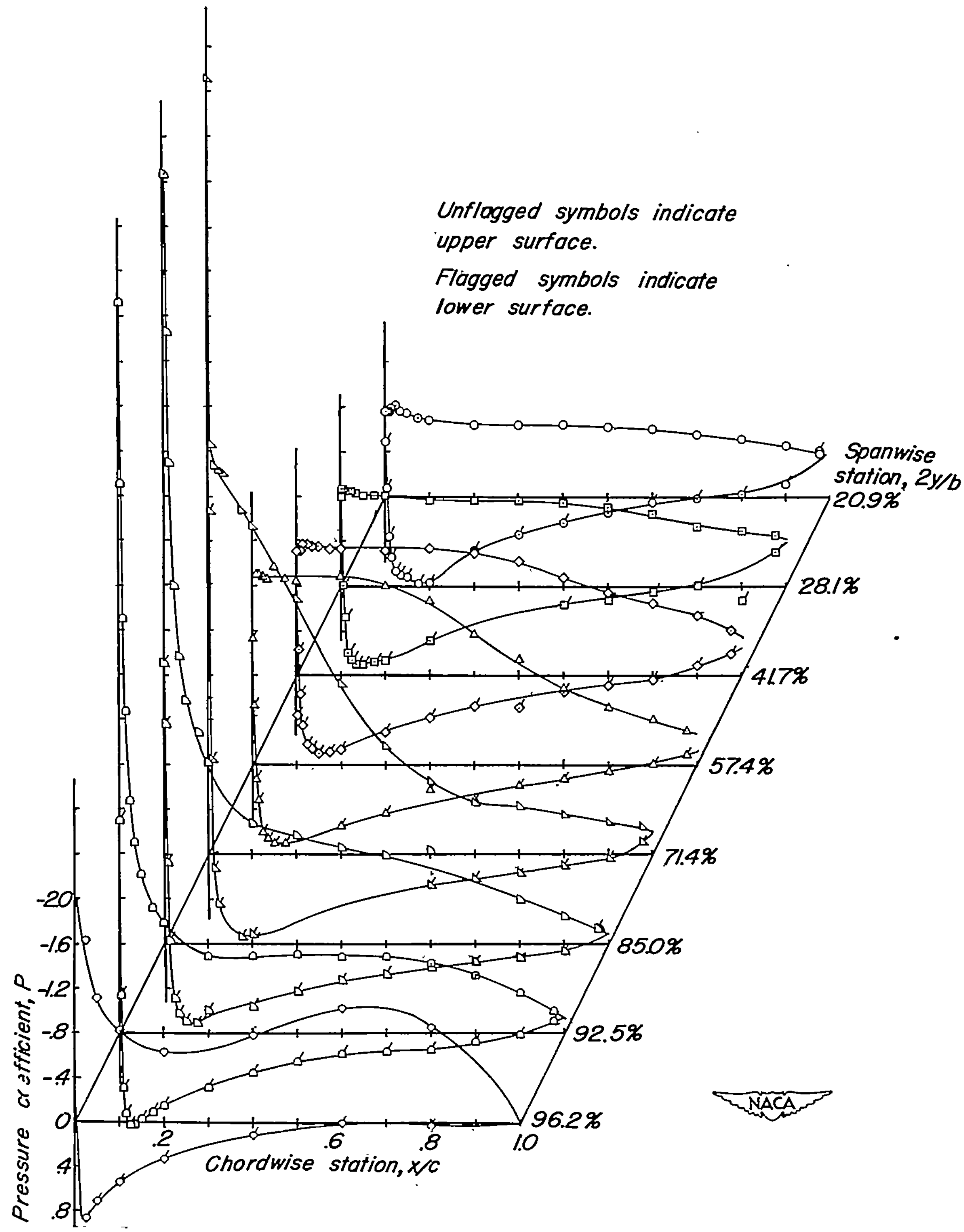
(f) α=16.6°

Figure 5.–Continued.



(g) $\alpha = 20.7°$

Figure 5.– Continued.



(h) α=24.8°

Figure 5.–Continued.

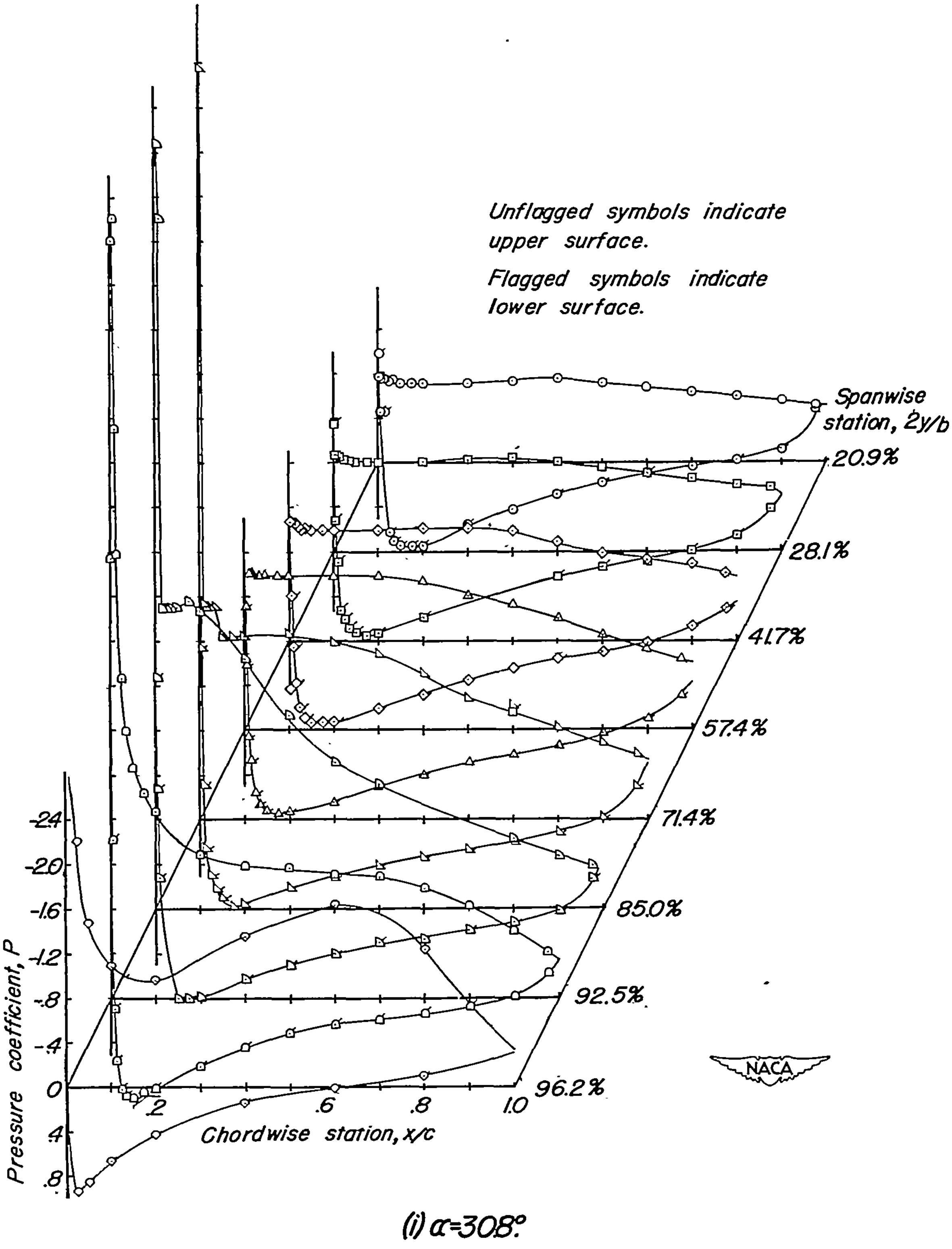


Figure 5.-Concluded.

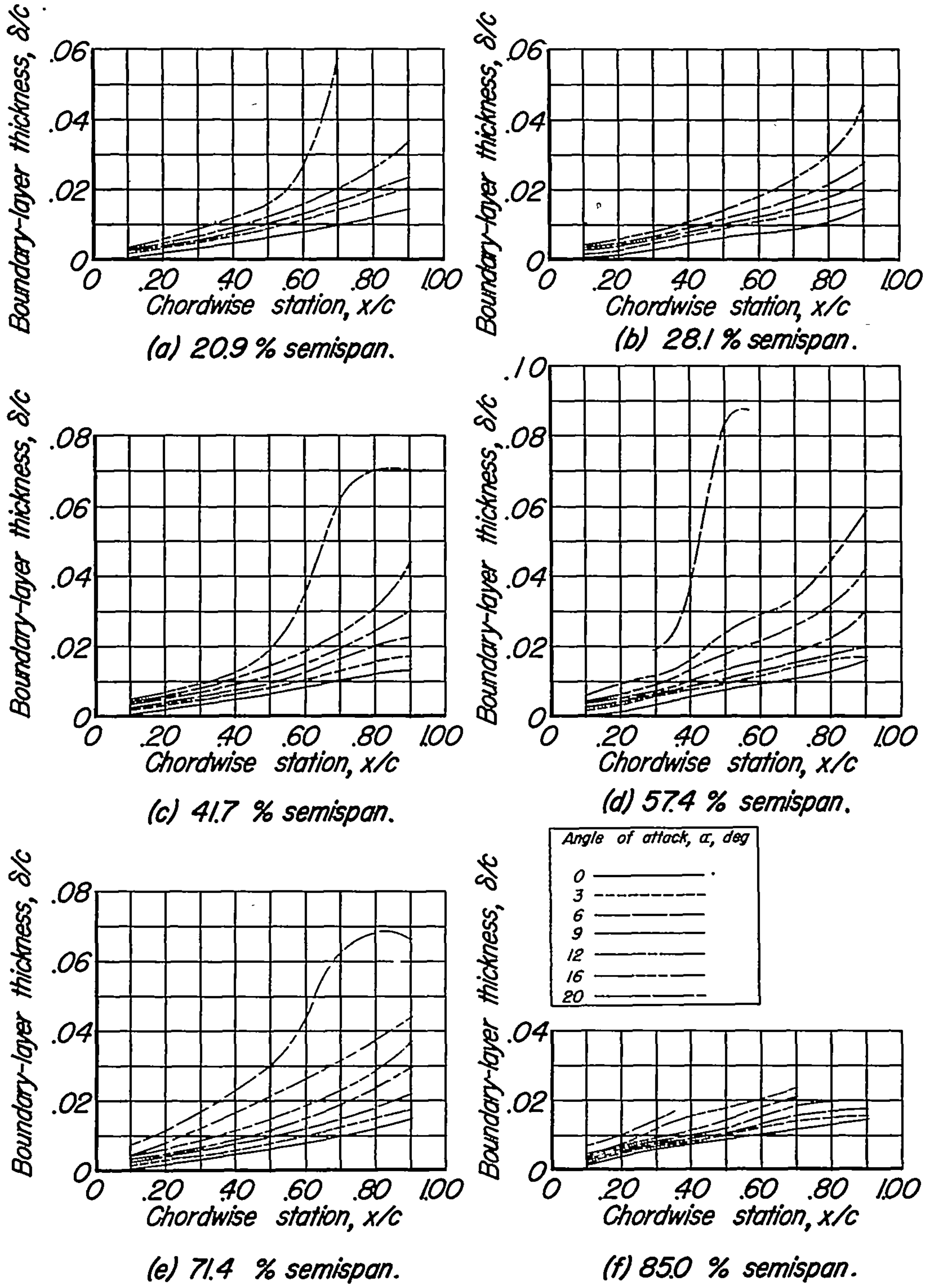


Figure 6.– Boundary-layer growth over six spanwise stations of the 45° swept-forward wing.

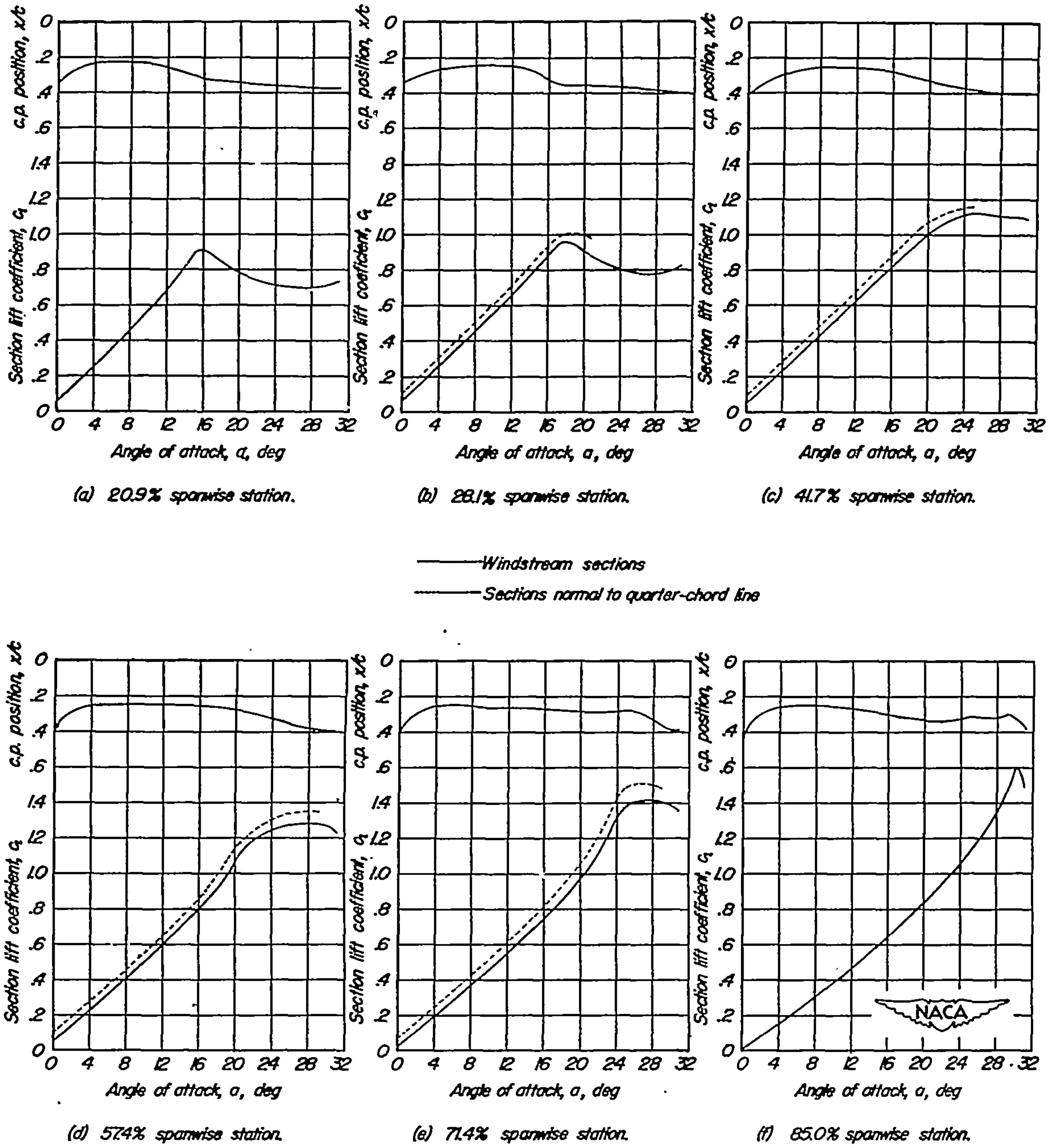


Figure 7.- Characteristics of sections at six spanwise stations of the 45° swept-forward wing.